bled from the compressor should bypass the fuel heat exchanger.

These objects are attained in an aircraft gas turbine engine in accordance with the invention by providing a turbine wherein the rotor disk bears a plurality of hollow, air-cooled turbine blades. Cooling air is bled from the high-pressure compressor and routed into a heat exchanger mounted at a location remote from the turbine, for example, on the outside of the compressor casing. Heat which has been introduced into the cooling air through the compression process is extracted within the heat exchanger by fuel which acts as a heat sink when routed through the heat exchanger and brought into heat exchange relationship with the cooling air, either directly or by way of an intermediate inert or nonflammable fluid medium. The use of inert or nonflammable fluid as the intermediate medium eliminates the possibility of fuel entering the cooling air flow path, which could result in the event of a leak in a heat exchanger where the fuel and cooling air are in direct heat exchange relationship.

The cooled cooling air is then directed from the heat exchanger, further compressed and then passed through the turbine rotor blades to provide improved cooling thereof. For higher coolant flows, fan bypass air can be used as a supplemental or alternative heat sink. The use of fuel as the heat sink results in a partially regenerative engine in which the heat removed from the compressed air is reintroduced more efficiently into the engine cycle.

Incorporation of the turbine blade cooling system in accordance with the invention into an aircraft or other gas turbine engine permits a reduction in the quantity of compressor air required for turbine rotor blade cooling and thus provides an improvement in engine performance. Conversely, an increase in blade life can be achieved by maintaining the original coolant flow rate but reducing the temperature of the coolant, with essentially no degradation in engine performance, or the turbine entry temperature can be increased to raise power output.

BRIEF DESCRIPTION OF THE DRAWINGS

These and other advantages of the invention will be better understood when the detailed description of the preferred embodiment of the invention is read in conjunction with the drawings, wherein:

FIG. 1 is a partial cross-sectional view of an aircraft gas turbine turbofan engine in accordance with a first preferred embodiment of the invention and illustrating schematically the relationship of various systems;

FIG. 2 is an enlarged, fragmentary, cross-sectional view depicting the internal construction of the pin or fin heat exchanger in accordance with the invention;

FIG. 3 is a block diagram illustrating schematically the relationship of various systems in accordance with a second preferred embodiment of the invention;

FIG. 4 is a front view of the coolant impeller and turbine disk in accordance with the preferred embodiments of the invention; and

FIG. 5 is a sectional view depicting the manner in which the coolant impeller of FIG. 4 is mounted.

DETAILED DESCRIPTION OF THE PREFERRED EMBODIMENTS

In FIG. an aircraft gas turbofan engine incorporating the invention is generally indicated by the numeral **10**. This engine generally comprises a core engine **12**, a fan assembly (not shown) including a stage of fan blades (not shown), and a fan turbine (not shown) which is interconnected to the fan assembly by rotatable shaft **16**. The core engine **12** includes an axial-flow high-pressure compressor **18** having a rotor **20** and a compressor casing **22** bearing a plurality of stators **24** interposed in alternating relationship with the stages of rotor **20**. Each stage of rotor **20** bears a plurality of radially directed, circumferentially distributed rotor blades **26** and each stator **24** bears a plurality of radially directed, circumferentially distributed stator guide vanes **28**.

Air enters the inlet (not shown) of and is initially compressed by the fan assembly. A first portion of this compressed air enters the fan bypass duct defined, in part, by core engine **12** and a circumscribing fan nacelle (not shown) and discharges through a fan duct **29** (only a portion of which is shown) and a fan nozzle (not shown). A second portion of the compressed air may be further compressed by a booster or other compressor and then enters inlet **30**, is further compressed by the axial-flow compressor **18** and then is discharged to a combustor **32**. In the combustor **32** the air is mixed with fuel. The fuel is supplied to fuel manifold **33** by means such as a fuel pump **34** and an engine fuel control **36** of a type well known in the art and responsive to pilot throttle inputs. The mixture is burned to provide high-energy combustion gases which drive a core engine turbine rotor **38**.

Core engine high-pressure turbine rotor **38** comprises a turbine disk **40** bearing a plurality of hollow turbine rotor blades **42** about its periphery. The turbine rotor **38** drives, in turn, the compressor rotor **20** through interconnecting shaft **44** in the usual manner of a gas turbine engine. A stationary row of turbine nozzle vanes **46** orients the flow into the rotating turbine rotor blades **42**.

The hot combustion gases then pass through and drive the fan turbine, which in turn drives the fan assembly. A propulsive force is thus obtained by the action of the fan assembly discharging air from the fan bypass duct through the fan nozzle and by the discharge of combustion gases from a core engine nozzle (not shown), the structure of which is well known in the art.

In accordance with a first preferred embodiment of the present invention, a turbine cooling system is provided which bleeds air from the high-pressure compressor **18**, transfers heat from that compressor bleed air to the fuel to be fed to the combustor **32**, and then supplies the cooled compressor bleed air to the cooling circuits (not shown) of the rotor blades **42** of the high-pressure turbine rotor **38**. The turbine cooling system generally includes an annular outlet manifold **48** for bleeding air from the high-pressure compressor **18**, a heat exchanger **50** for transferring heat by conduction from the compressor bleed air to the fuel being fed to the combustor **32**, an annular inlet manifold **52** for circumferentially distributing the cooled compressor bleed air returned to the core engine from the heat exchanger, and an impeller **54** for further compressing and feeding the cooling air to the hollow turbine rotor blades **42**.

In accordance with the invention, the compressor bleed air is extracted through a plurality of openings **56** which communicate with outlet manifold **48**. For the purpose of illustration only, FIG. shows the pressurized air being extracted behind the fourth-stage rotor, although the air may in the alternative be extracted further downstream or further upstream. The precise point of extraction will be a function of the amount of pressur-